

REPORT DOCUMENTATION PAGE				Form Approved OMB No. 0704-0188	
Public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing this collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden to Department of Defense, Washington Headquarters Services, Directorate for Information Operations and Reports (0704-0188), 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202-4302. Respondents should be aware that notwithstanding any other provision of law, no person shall be subject to any penalty for failing to comply with a collection of information if it does not display a currently valid OMB control number. PLEASE DO NOT RETURN YOUR FORM TO THE ABOVE ADDRESS.					
1. REPORT DATE (DD-MM-YYYY) 25-06-2009		2. REPORT TYPE Technical Paper		3. DATES COVERED (From - To)	
4. TITLE AND SUBTITLE A Review of High Thrust, High Delta-V Options for Microsatellite Missions				5a. CONTRACT NUMBER	
				5b. GRANT NUMBER	
				5c. PROGRAM ELEMENT NUMBER	
6. AUTHOR(S) David B. Scharfe (ERC); Andrew Ketsdever (AFRL/RZSA)				5d. PROJECT NUMBER	
				5f. WORK UNIT NUMBER 50260542	
7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES) Air Force Research Laboratory (AFMC) AFRL/RZSA 10 E. Saturn Blvd. Edwards AFB CA 93524-7680				8. PERFORMING ORGANIZATION REPORT NUMBER AFRL-RZ-ED-TP-2009-267	
9. SPONSORING / MONITORING AGENCY NAME(S) AND ADDRESS(ES) Air Force Research Laboratory (AFMC) AFRL/RZS 5 Pollux Drive Edwards AFB CA 93524-7048				10. SPONSOR/MONITOR'S ACRONYM(S)	
				11. SPONSOR/MONITOR'S NUMBER(S) AFRL-RZ-ED-TP-2009-267	
12. DISTRIBUTION / AVAILABILITY STATEMENT Approved for public release; distribution unlimited (PA #09322).					
13. SUPPLEMENTARY NOTES For presentation at the 45 th AIAA Joint Propulsion Conference & Exhibit, Denver, CO, 2-5 August 2009.					
14. ABSTRACT Microsatellites have been suggested as a means of enhancing a variety of proposed space missions, ranging from low-Earth-orbit to solar-system exploration. With improvements in propulsion technology geared toward microsatellites, the ultimate delta-V (ΔV) capabilities of some microsatellite systems are now in the range of several km/s, opening the doors to a variety of high ΔV , fast response scenarios. This paper provides a brief overview of propulsion technologies currently available for microsatellites, and an evaluation of each technology for potential use in a demanding mission. The sample mission is that of a microsatellite inspector which, starting in a 200 km parking orbit, must be diverted to rendezvous with another satellite in orbit at a different altitude and inclination. It is found that existing bipropellant microrocket designs provide a high thrust value, combined with a 300 s specific impulse, allowing for response times of only a few hours for such an inspector mission with ΔV requirements over 1 km/s. Miniaturized electrostatic thrusters provide the largest ultimate ΔV capability, approaching 10 km/s, but with a very low thrust level and therefore a response time capability of several months. Newly developed micro-solar thermal systems fill in the middle ground for these two options, providing the moderate thrust levels and specific impulse values necessary for a response time on the order of one day and a ΔV of several km/s.					
15. SUBJECT TERMS					
16. SECURITY CLASSIFICATION OF:			17. LIMITATION OF ABSTRACT	18. NUMBER OF PAGES	19a. NAME OF RESPONSIBLE PERSON
a. REPORT	b. ABSTRACT	c. THIS PAGE			Dr. Andrew Ketsdever
Unclassified	Unclassified	Unclassified	SAR	15	19b. TELEPHONE NUMBER (include area code) N/A

A Review of High Thrust, High Delta-V Options for Microsatellite Missions

David B. Scharfe*

ERC Incorporated, Edwards AFB, CA, 93524

Andrew D. Ketsdever†

Air Force Research Laboratory, Edwards AFB, CA, 93524

Microsatellites have been suggested as a means of enhancing a variety of proposed space missions, ranging from low-Earth-orbit to solar-system exploration. With improvements in propulsion technology geared toward microsatellites, the ultimate delta-V (ΔV) capabilities of some microsatellite systems are now in the range of several km/s, opening the doors to a variety of high ΔV , fast response scenarios. This paper provides a brief overview of propulsion technologies currently available for microsatellites, and an evaluation of each technology for potential use in a demanding mission. The sample mission is that of a microsatellite inspector which, starting in a 200 km parking orbit, must be diverted to rendezvous with another satellite in orbit at a different altitude and inclination. It is found that existing bipropellant microrocket designs provide a high thrust value, combined with a 300 s specific impulse, allowing for response times of only a few hours for such an inspector mission with ΔV requirements over 1 km/s. Miniaturized electrostatic thrusters provide the largest ultimate ΔV capability, approaching 10 km/s, but with a very low thrust level and therefore a response time capability of several months. Newly developed micro-solar thermal systems fill in the middle ground of these two options, providing the moderate thrust levels and specific impulse values necessary for a response time on the order of one day and a ΔV of several km/s.

I. Introduction

LIGHT-WEIGHT (100 kg class and smaller) microsatellites, combined with miniaturized spacecraft components, are a well-established technology proven to reduce the costs and enhance the capabilities of certain space missions. Though the capabilities of microsatellites are traditionally more limited than those of their larger counterparts, the relatively small mass of microsatellites could allow for drastically reduced launch costs; reduced development times for microsatellites may also result in the use of more modern technology, which can enhance capabilities and mitigate some of the compromises made to reduce system mass.¹ Additionally, the concept of producing several similar or identical microsatellites provides another avenue for targeting the needs of specific missions; such a group of microsatellites could be flown in a cluster to replace a single large satellite, or flown separately to perform various aspects of a mission requiring presence at multiple locations or at various times. Further, if a satellite were to become damaged while in orbit, replacing a single microsatellite is fairly trivial when compared to replacing an entire, traditionally-sized satellite: a single, smaller satellite can be constructed and launched on shorter notice, and likely with decreased expense, when compared to the process required for a full-sized satellite.

The above-listed advantages of microsatellite missions are well known, but as-yet, microsatellite capabilities in terms of the ultimate velocity increment (ΔV) available for station keeping, orbit transfers, and other maneuvers have been viewed as somewhat limited. However, newer technologies, by way of advanced micro-chemical and micro-electric propulsion systems, present the opportunity to remove that limitation,

*Senior Scientist, Advanced Propulsion Concepts, 10 E. Saturn Blvd., Edwards AFB, CA 93524, AIAA Member.

†Program Manager, Advanced Propulsion Concepts, 10 E. Saturn Blvd., Edwards AFB, CA 93524, AIAA Assoc. Fellow.

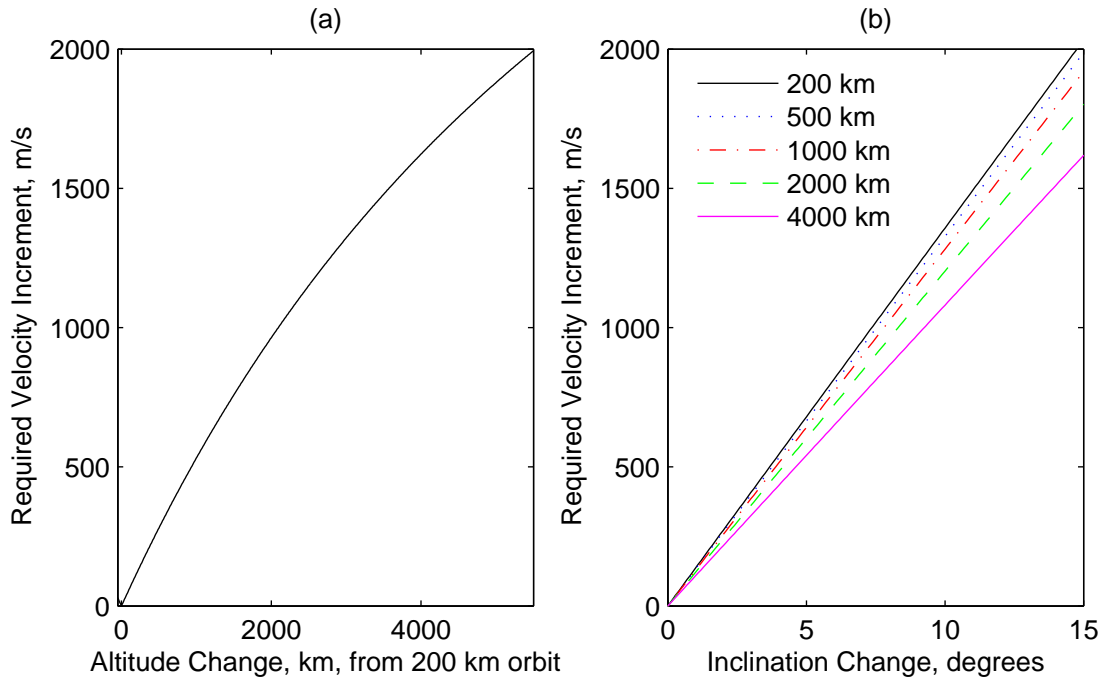


Figure 1. (A): Required velocity increment to alter the altitude of the initial 200 km circular orbit. A Hohmann transfer is assumed. (B): ΔV required for inclination change at various altitudes.

providing a combination of high thrust, high specific impulse, and high propellant throughput suitable for total velocity changes on the order of kilometers per second. Proposed microsatellite missions now include transfers from low-Earth-orbit (LEO) to geostationary orbit (GEO), transfers to lunar orbits, and even transfers beyond Earth orbit to explore other planets and various Near-Earth-Objects (NEO's).²⁻¹²

It is the goal of this paper, therefore, to explore the feasibility of various technologies for achieving the goals of a novel, high- ΔV , fast-response mission using microsatellite technology. The proposed mission is that of a microsatellite inspector that can, on short notice, be diverted from its orbit to rendezvous with another satellite. The purpose of this rendezvous may range from a simple close-range optical inspection or analysis of the destination satellite, to physical contact for purposes of increasing the target satellite's mission life via maintenance, refueling, or other such operations.^{11,13} Although many such inspector satellites could potentially be launched into varying initial orbits to yield flexibility and redundancy, and to limit the ΔV requirements on any one inspection mission, ΔV requirements are expected to be on the order of 1 km/s. In the interest of response time, thrust requirements will be placed at 10 mN or higher, though thrust levels over 1 N are ideal to ensure that required orbit alterations can be achieved in under one day.

II. Generalized Calculations

A microsatellite inspector, like that described above, would require a significant velocity increment to achieve the inclination change and altitude alteration necessary to rendezvous with another satellite (or several other satellites) in LEO. Additional ΔV requirements for slight rephasing of the orbit, or for a precise rendezvous with a target satellite, are neglected in this analysis.

Assuming an initial circular orbit at an altitude of 200 km, the ΔV required to alter the orbit altitude to various degrees (via Hohmann transfer) is illustrated in the plot provided in Figure 1(a). Limiting the microsatellite to a maximum velocity increment of 1.5 km/s, an increase in altitude of 3500 km (to a 3700 km orbit altitude) could potentially be achieved.

Likewise, the ΔV 's required for various levels of inclination change at the altitudes the microsatellite might achieve are illustrated in Figure 1(b). Again limiting the microsatellite to a maximum velocity increment of

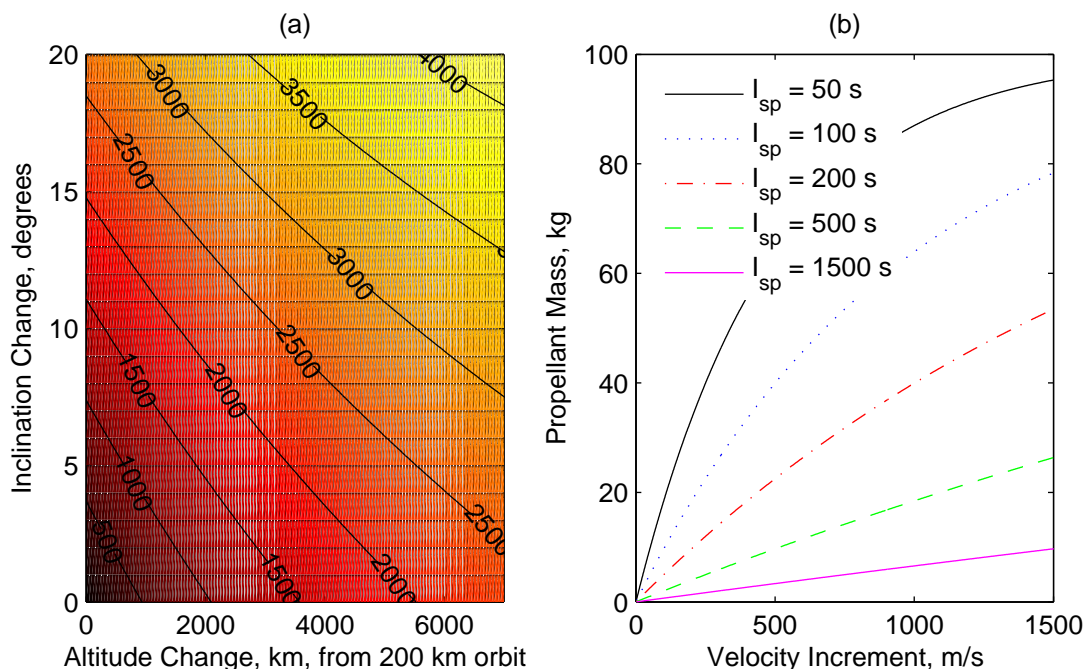


Figure 2. (A): Color map and contours indicating the required ΔV values required to achieve a given combination of altitude increase and inclination change. Units of the ΔV are m/s. (B): Required propellant mass for a 100 kg microsatellite to achieve a desired velocity increment utilizing thrusters with various I_{sp} values.

1.5 km/s, and keeping the altitude at 200 km, an inclination change of 11° is possible.

Combining the calculations used to produce the plots in Figure 1, and assuming the inclination change is performed after circularizing the orbit at the higher altitude, Figure 2(a) was produced. Combined altitude and inclination changes could be optimized to reduce the total ΔV required, but performing the calculation in the manner described will provide an upper limit to the total required velocity increment, and allow for smaller, less certain, ΔV components such as minor orbit rephasing, attitude control, and precise rendezvous approach. Again limiting the craft to a total ΔV of 1.5 km/s, one can, for example, estimate that raising the orbit altitude to 2000 km (an 1800 km increase in altitude) requires 880 m/s ΔV , allowing for just over 5° of inclination change to the final orbit.

The required propellant mass for a 100 kg microsatellite, as a function of desired ΔV and thruster I_{sp} , is illustrated in Figure 2(b); calculations were performed using the idealized rocket equation. Whereas a higher I_{sp} thruster requires less propellant, it would also generally require an electric propulsion system with a larger power supply and solar array, impinging on some (or all) of the mass benefit associated with reduced propellant consumption. Note also that in such systems, a higher specific impulse typically indicates reduced thrust, which will affect the transfer time capabilities of the microsatellite. Nonetheless, with any of the I_{sp} ranges indicated in the plot, and achieved by various miniaturized thruster technologies, a microsatellite with a 1.5 km/s (or greater) ΔV capability is technically feasible. However, if a propellant fraction of 80% or less is required to allow for a useful payload capacity, then only those systems with I_{sp} values in excess of 100 s will likely be suitable for the proposed mission.

III. Microsatellite Propulsion Technologies

A previous, exhaustive review of microsatellite propulsion technologies was completed by Mueller a decade ago.^{14,15} In the discussion to follow, we seek to survey the more recent developments in the field, while focusing on the high thrust and high ΔV needs of a rendezvous mission.

Multiple microsatellite propulsion technologies that may be suitable or adaptable to a satellite-rendezvous mission have been described recently in the literature. These technologies range from arrays of solid-

Table 1. Summary of propulsion technologies available for microsattellites. Data has been omitted where it is unavailable in the literature (spaces left blank) or is effectively irrelevant (marked with -). Note that some systems such as solar thermal and electric rockets require additional mass for power systems, which is not included in the thruster mass.

Thruster Type	References	Thrust	I_{sp} [s]	Power [W]	Thruster Mass
Hall/Ion	11, 22–24, 26–34	0.4–20 mN	300–3700	14–300	≤ 1 kg
FEED/colloid	11, 25, 26	0.1 μ N–1.5 mN	450–9000	1–100	0.1–1 kg
Electromagnetic	11, 22, 26, 35	0.03–2 mN	200–4000	≤ 10	0.06–0.5 kg
Electrothermal	6, 11, 26, 36–39	≤ 220 mN	50–250	3–300	0.1–1 kg
Cold Gas	16, 26, 40	0.5 mN–3 N	40–80	–	0.01–1 kg
Monopropellant	4, 11, 25, 26, 41–43	1 μ N–1.5 N	100–230	≤ 6	0.01–0.5 kg
Bipropellant	4, 6, 11, 16, 19, 25, 26, 44	1 μ N–45 N	100–320	≤ 6	0.01–0.5 kg
Decomposing Solid	45		230		
Laser Micro. (ablation)	18	1 μ N	100–300	2	
Laser Micro. (ignition)	18	1–10 mN	37–100	–	
Laser Plasma	3, 46	0.1–1 mN	500–1000	2	≤ 1 kg
Hollow Cathode	47	1 μ N–10 mN	50–1200	5–1000	
Solar Thermal	4, 8, 9, 20, 48–50	56 mN – 1 N	200–1100	–	≤ 10 kg

propellant digital microthrusters^{16–18} through bipropellant microrockets with 10’s of newtons of thrust and I_{sp} values over 300 s.¹⁹ Non-chemical technologies include solar-thermal propulsion mechanisms offering several hundred seconds of I_{sp} while producing newtons of thrust,^{9,10,20} and electric propulsion technologies for microsattellites that offer low thrust but up to several thousand second specific impulse values.^{7,21–25}

A partial survey of existing microsattellite propulsion technologies was completed by Stein, et al. in 2008.²⁶ An updated and augmented version of this compilation, including specifications of more advanced and higher-thrust miniaturized designs, is provided in Table 1.

When analyzing the details of Table 1, it is important to note that the specifications indicate traits of individual thrusters, whereas many of the technologies listed lend themselves well to satellite designs with multiple identical thrusters working in unison. Small-size, low-power thrusters are ideal for this type of operation, and a microsattellite designed with multiple small propulsive units of any of the types listed in Table 1 (except perhaps the higher-powered electric rockets) can be imagined. This is a common assumption, for example, in the design of microelectromechanical (MEMS) based bipropellant chips.^{6,16,19} This strategy of design lends itself to an enhancement over the performance (thrust) of a single thruster, and also yields increased redundancy (and therefore reliability) relative to a single, larger thruster. As an example of such a configuration, a clear illustration of this concept is provided by Marcu, et al.,¹⁹ and it is reproduced here as Figure 3.

Table 2. Propellant mass fraction required to achieve a 1.5 km/s velocity increment using select microthruster technologies. The I_{sp} values assumed for each technology are typical, selected from within the ranges provided in Table 1.

Thruster Type	Assumed I_{sp} [s]	Propellant Fraction [%]
Bipropellant	320	38
Monopropellant	230	49
Cold Gas	60	92
Solar Thermal (H ₂)	1000	14
Solar Thermal (NH ₃)	400	32
Electrothermal	100	78
Electrostatic (Hall)	1000	14
Electrostatic (Ion)	3000	5

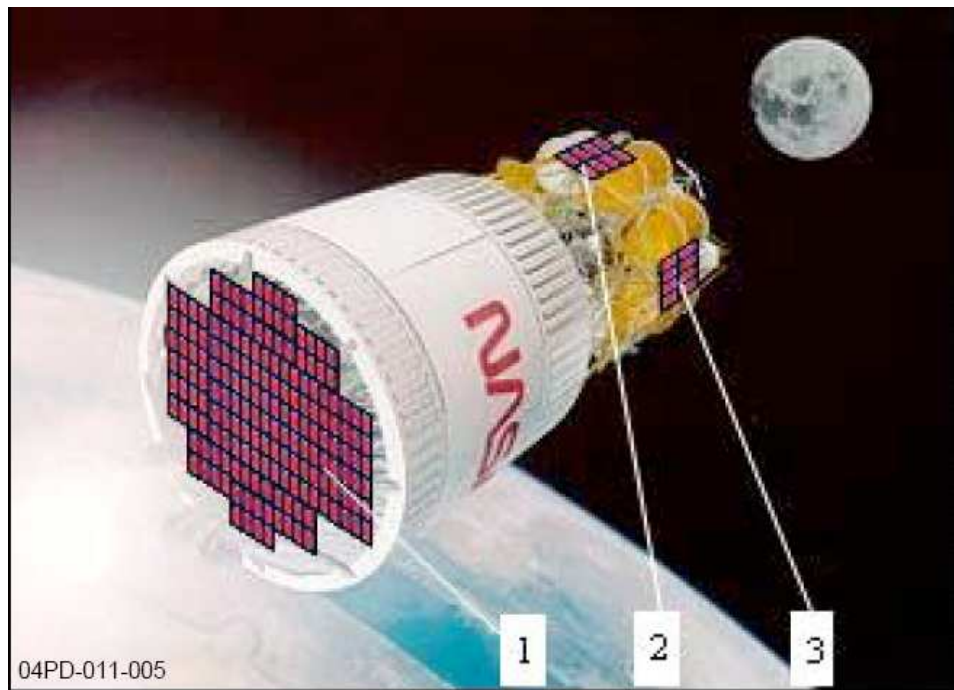


Figure 3. Conceptual design illustrating arrays of microrockets set up for main propulsion (1), RCS (2), and ACS (3). This design allows for enhanced and highly variable thrust, and increased redundancy for reliability purposes. Figure taken from Marcu, et al.¹⁹

It is also noted here that the Hollow Cathode Thruster⁴⁷ design operates in multiple modes. At higher powers, the I_{sp} and thrust are at the top of the ranges listed in Table 1; at lower powers, both the I_{sp} and thrust are correspondingly small.

In cataloguing these technologies, novel methods of altering a spacecraft's trajectory such as the use of a "space tug"⁶ or tether^{12, 51} system have not been included. It is assumed that these systems do not provide the flexibility required for a microsatellite inspector type mission.

IV. Miniaturized Thruster Evaluation

In analyzing the likely needs of a high ΔV mission such as that of a microsatellite inspector, relatively high thrust values become important to enable a reasonable response time. To a first-order approximation, and assuming a relatively large 1 km/s velocity increment, achieving a response time on the order of 1 day requires thrust of approximately 1 N for a satellite with an initial mass of 100 kg. Such a fast (or faster) response time may be necessary in the case of inspecting a newly-launched satellite for damage or functionality. However, in the case of a repair or service (i.e.: refueling) mission, pre-planning may allow for a more extended approach to the target satellite. However, response time on the order of weeks or months (requiring 10 mN of thrust or more), rather than years or decades, are still expected to be necessary.

Therefore, technologies listed in Table 1 with thrust of only a few millinewtons or less are likely not suitable for the type of mission under investigation here. For this reason, low thrust electrostatic devices such as FEEP and colloid thrusters, electromagnetic devices such as micro-pulsed plasma thrusters (μ PPTs) and vacuum arc thrusters (VATs), and laser-based thruster systems will be excluded from the discussion to follow.

Additionally, power consumption on a microsatellite is of significant concern. Taking the 1 W/kg order of magnitude estimate of satellite power used by Mueller,¹⁵ a 100 kg microsatellite can only provide up to 100 W of continuous power. Thruster technologies that require much more than this figure are likely not suitable for this mission design.

Finally, the overall mass of the propulsion system, including the thruster itself and any required pumps,

power systems, high-pressure tanks, or other such components, is of crucial concern for a mass-limited microsatellite. With a high- I_{sp} thruster that can achieve a large ΔV using relatively little propellant, a system weighing up to several kilograms may be acceptable; for a system requiring more propellant to achieve the same ΔV , the system mass will be more critical to allow for the extra required propellant mass. To illustrate the necessity of this trade-off, the propellant mass fractions required for select microsatellite thruster technologies to achieve a 1.5 km/s ΔV are listed in Table 2. Note, for example, that achieving a 1.5 km/s change in velocity with a cold gas jet may require over 92 kg of propellant on a 100 kg microsatellite.

Based on the evaluation metrics discussed above, the miniaturized thruster technologies with the most potential utility in a microsatellite inspector type mission are: Bipropellant, Monopropellant, Cold Gas, Solar Thermal, Electrothermal, and Electrostatic (Hall/Ion) thrusters. These technologies will be further explored below. It is noted here that while the term “microthruster” is often used to denote a thruster which produces thrust in the sub-millinewton range, the thrusters under consideration here are notable for their small size and power consumption, but are intended to produce large enough thrust to impart a significant velocity increment to the microsatellite in a short amount of time.

IV.A. Bipropellant Microrockets

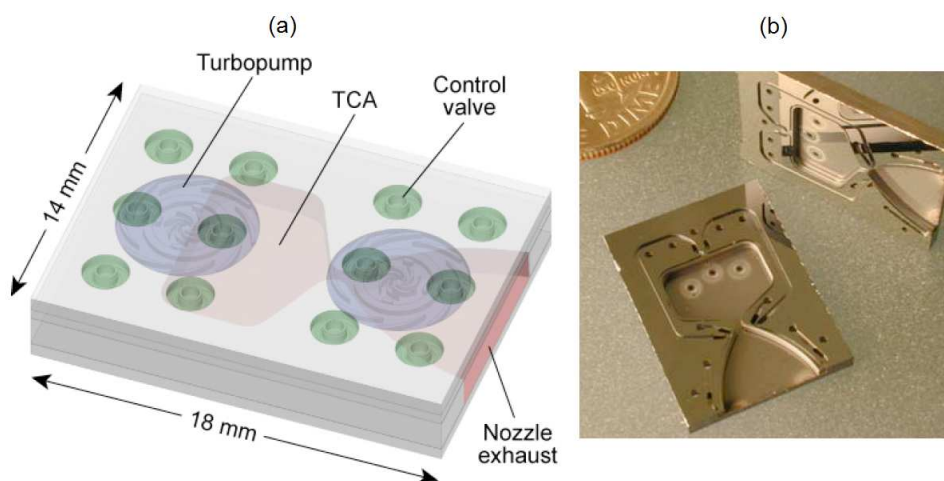


Figure 4. (A): Design of a 3–10 lb thrust MEMS bipropellant rocket. (B): Photograph of MEMS-fabricated bipropellant rocket nozzle; plumbing and cooling detail is evident. Figures taken from Marcu, et al.¹⁹

Bipropellant microrockets are advanced MEMS devices, effectively integrating the necessary valves, pumps, combustion chamber, and nozzle onto a single microfabricated chip or assembly. Perhaps the most notable of the present options available in the literature is the design proposed by Marcu, et al.,¹⁹ which is estimated to produce up to 10 lb (45 N) of thrust per chip at an I_{sp} of 300 s. The thrust-to-weight ratio of the chip is roughly 1000:1, indicating that the thruster mass, including turbopumps, valves, sensors, the combustion chamber, and the nozzle, weighs in at well under 100 grams. Further, the use of the turbopumps to pressurize the fuel is suggested as a means to save system mass by eliminating the need for a high-pressure propellant storage tank.²⁵ Each microrocket chip is fabricated from several layers of silicon and has a total volume of less than 2 cm³. This thruster design is illustrated in Figure 4.

The cost per newton of thrust for a chip-based propulsion system is expected to be a factor of 5–10 less than that of a traditional upper stage bipropellant engine.¹⁹ These chip-based engines consume very little power, and can be operated in clusters (see Figure 3) to enhance mission flexibility and reliability. Individual chips can still be throttled to vary thrust, and differing numbers of chips can be operated to further vary the thrust output. The chip-based bipropellant engines can still operate at hundreds of atmospheres of internal pressure with I_{sp} values similar to those of their full-scale counterparts, so the performance penalty relative to full-sized conventional rockets is believed to be minimal.

Other, similar miniaturized bipropellant thruster technologies have been presented,^{4,6,11,16,25} all providing roughly 300 s of I_{sp} with thrust levels well above 1 N, and consuming only a few watts of power.

While much smaller bipropellant engines have also been designed, offering millinewtons of thrust,^{11,44} large efficiency losses due to heat loss and resistance to the flow are likely to occur in these very small bipropellant engines (i.e.: true microthrusters producing millinewtons or micronewtons of thrust). No such losses seem to be apparent at thrust levels greater than 1 N; the exhaust velocity (I_{sp}) of these MEMS microrockets should be equivalent to that of a full-sized thruster operating with the same propellants and combustion cycle.¹⁹

With multi-newton thrust values available to a microsatellite, fast response time for even a large ΔV is quite feasible. At 45 N of thrust, a ΔV of 1 km/s could be achieved in well under an hour for a 100 kg microsatellite. Clustering of several of these MEMS microrockets can be used to further enhance the thrust and response time, possibly achieving the thrust level required to approximate an instantaneous ΔV for a Hohmann-type orbit transfer.

Additionally, as can be seen via Figure 2(b), with an I_{sp} value of 300 s, well under 50 kg of propellant would be required to provide a 1.5 km/s ΔV , leaving enough mass budget on board the microsatellite for purposes such as payload mass or other satellite subsystems.

IV.B. Monopropellant Microrockets

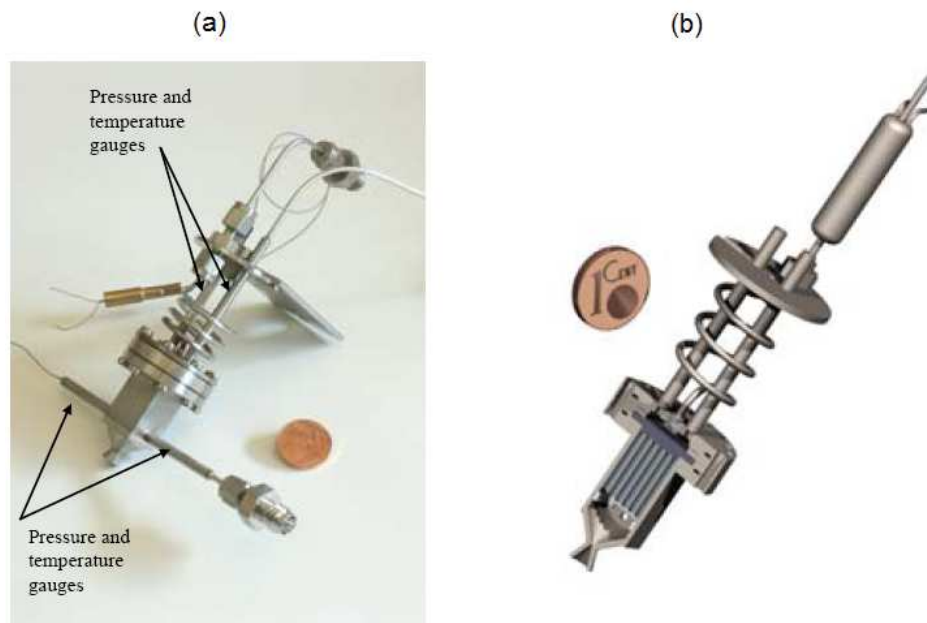


Figure 5. (A): Laboratory model of a monopropellant thruster. (B): Cross section thereof showing the internal assembly with the catalyst. Figures taken from Scharlemann, et al.²⁵

Miniaturized monopropellant rockets, like that illustrated in Figure 5, are very similar to their bipropellant counterparts, but with a simpler design due to the need for only a single propellant feed. Monopropellant designs also generally require a catalyst bed to ignite the propellant,^{11,25,41–43} which is typically hydrazine or hydrogen peroxide. Additionally, the development of monopropellant designs for main propulsion systems on microsatellites benefits from the wide-scale use of small hydrazine thrusters for attitude control on larger satellites.⁴¹

As with the bipropellant engines, the propellant can be stored in a pressurized tank (with the thruster operating in blowdown mode), or a miniaturized turbopump could be used to pressurize the fuel. Additionally, much like the bipropellant engines discussed above, small monopropellant thrusters are compact enough and consume little power so that they can be clustered to increase thrust, redundancy, and flexibility.

The miniaturized monopropellant thrusters catalogued in Table 1 have maximum thrust values only on the order of 1 N,^{11,25,41–43} which is notably lower than the bipropellant thrusters. However, this deficiency is likely non-technological and merely an artifact of individual engineering goals or mission requirements. Many of the proposed thrusters are, in fact, intended for use on nanosatellites 10 kg or smaller. In many

cases, MEMS fabricated chemical propulsion systems are being developed due to the effective miniaturization of thruster designs using this technique.⁴³ Combustion research plays a critical role in the success of these microscale thrusters with researchers attempting to design efficient, high temperature chemical propulsion systems.⁵² The effects of miniaturization on the fluid mechanics, heat transfer, and combustion characteristics involved in micro-chemical devices will act to limit the efficiency of these systems. Research is required to investigate the effects of viscous boundary layers, microscale heat transfer, and large species gradients in order to improve the performance of micro-chemical propulsion systems.

A monopropellant microrocket would be simpler than a bipropellant system, requiring only a single fuel pump and one propellant tank. Whereas the monopropellant systems have notably lower I_{sp} values (230 s vs. 300 s for bipropellant engines), the simplicity and potential mass savings of such a system (only one propellant tank, etc. required) may outweigh this performance deficit. A propellant mass of 50 kg could still provide 1.5 km/s of ΔV , allowing for a highly capable microsatellite. However, some researchers have run into lifetime issues with the catalyst bed required to ignite the monopropellant, noting a decrease in reactivity after several kilograms of propellant throughput;⁴² this issue should be overcome via design optimizations.

IV.C. Miniature Cold Gas Thrusters

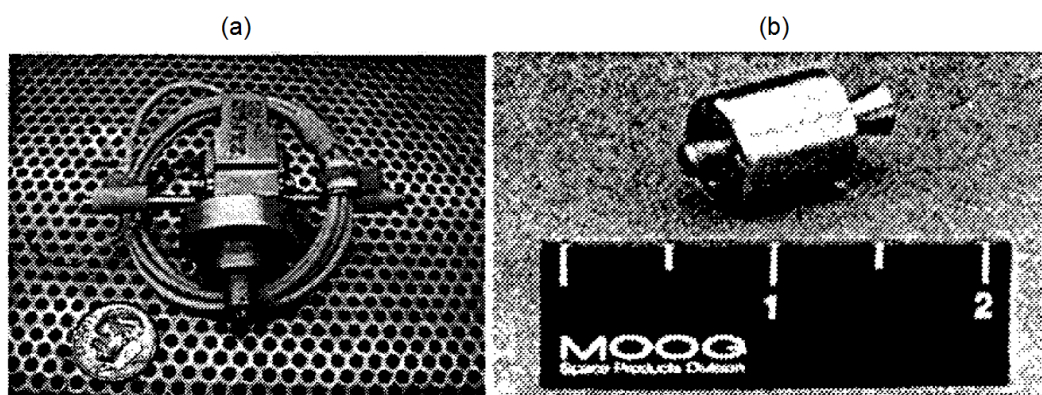


Figure 6. (A): 80 g Moog model 50X802 gas regulator assembly. (B): 3.5 N thrust, 22 g Moog model 58-118 SAFER thruster, including internal valves. Figures taken from Bzibziak.⁴⁰

Recent work in cold gas thrusters suitable for microsatellites, especially that by Moog Inc.,⁴⁰ has shown thrust levels of several newtons from thrusters weighing only a few grams. As an entire assembly, including miniaturized pressure regulators, valves, and thruster nozzles, but neglecting the high-pressure propellant tank, the system mass would be under 1 kg. All components have been designed to operate with high pressures and very low leak rates. Samples of these miniaturized components are illustrated in Figure 6.

Whereas the thrust level of such a cold gas microthruster is suitable for a relatively fast response, the I_{sp} value of a cold gas thruster is generally only 40-80 s, which significantly hampers the performance. Achieving a 1.5 km/s ΔV with such a cold gas thruster would require a propellant mass fraction of over 80%, leaving little mass budget available for other purposes.

It is noteworthy, however, that these thrusters from Moog Inc.⁴⁰ were tested running nitrogen and xenon as the propellant gas; switching to a lighter propellant (such as hydrogen) would significantly increase the exhaust velocity (I_{sp}), but would also likely increase the mass of the tank required to store the propellant. With a similar volume flow rate, a lighter propellant gas would also result in decreased thrust.

IV.D. Solar Thermal Thrusters

Solar thermal thrusters, which concentrate solar rays to heat a compressed gas propellant, offer a step up from cold gas microthrusters. Whereas such designs require additional mass in the form of solar collectors, the I_{sp} can reach over 1000 s, with up to 1 N of thrust, operating on hydrogen propellant;^{4,9} heavier propellants such as ammonia can still produce I_{sp} values of 300–400 s,^{4,9,20,50} rivaling the capabilities of bipropellant engines in a very robust system. Two configurations, utilizing fiber optic lines to direct the collected solar

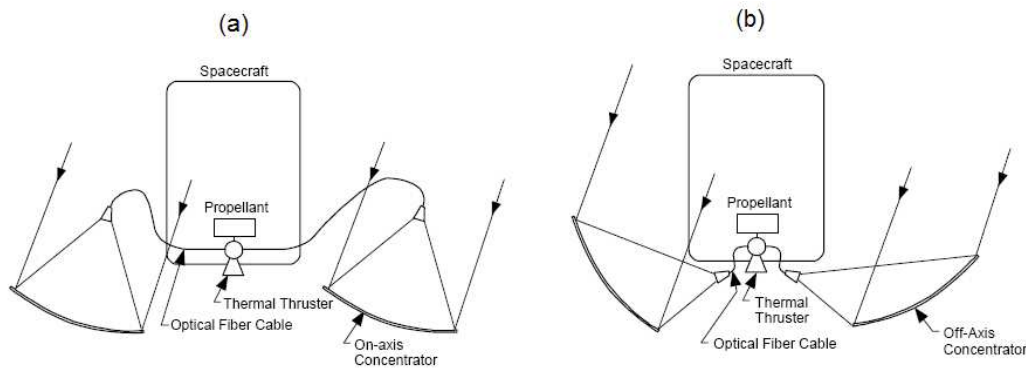


Figure 7. Basic schematics of fiber-coupled solar thermal propulsion systems. (A): On-axis solar collection. (B): Off-axis collection. Figures taken from Nakamura, et al.⁹

flux, are illustrated in Figure 7; the solar flux can also be reflected directly onto the propellant heating chamber.

Whereas it may seem that a solar thermal propulsion system would have strict limitations due to times of eclipse or strict thrust-vector pointing requirements based on the orientation of sun, this is not necessarily the case. Solar thermal systems have been designed with thermal-storage capabilities, allowing for augmented thrust (beyond that of a cold gas jet) in times of eclipse.^{49,50} Further, thrust-vector pointing relative to incoming sunlight can be varied by changing the orientation of solar concentrator arrays; additional flexibility in this regard can be offered via fiber optic cables used to pass sunlight from the concentrator focal point to the propellant heating cavity.^{9,49,50}

The mass of the collection system can be fairly large, possibly requiring several square meters of solar concentrator and a significant mass of fiber optic cables to carry the equivalent intensity of several thousand suns.⁹ A system mass of over 20 kg to achieve 1 N of thrust and 1000 s of I_{sp} on hydrogen is a possibility, but switching to (heavier) ammonia propellant at a similar thrust level and 400 s I_{sp} would significantly decrease the mass of the solar thermal propulsion system to roughly 7 kg, excluding the propellant.⁹ Additionally, this system design assumes several kilograms of fiber optic lines and couplings; reflecting sunlight directly onto the propellant heating zone could eliminate this mass, albeit at the cost of some system flexibility.

Even with a 20 kg solar thermal system, however, at an I_{sp} of 1000 s, only 20 kg of propellant would be required to achieve a 1.5 km/s ΔV , resulting in an entire propulsion system, including propellant, with a mass less than 50% that of a highly-capable microsatellite. With a 7–8 kg ammonia system with a 300–400 s I_{sp} , 40–50 kg of propellant would be required for such a ΔV , resulting in a similar fraction of the satellite mass being taken up by the propulsion system. At 1 N of thrust, this ΔV would be achievable in under one day, providing a reasonable response time for a microsatellite inspector or rendezvous type mission.

IV.E. Electrothermal Microthrusters

Electrothermal rockets, such as arcjets and resistojets, utilize electric power to heat a propellant gas. For microsatellite purposes, the amount of power that can be supplied (and therefore the maximum optimal gas flow rate) is limited by the power bus of the satellite; for a 100 kg satellite, this limit is likely around 100 W of continuous power.¹⁵

Small resistojets, like that illustrated in Figure 8, have been shown to operate on a wide variety of propellants, including ammonia, butane, hydrogen, nitrogen, and noble gases.³⁸ In the microsatellite power range, small resistojets producing up to 220 mN of thrust operating on up to 80 W of augmenting power have been demonstrated.¹¹ Two examples include a 13.7 W resistojet operating on butane propellant to produce 100 mN of thrust at 100 s of I_{sp} ^{11,38} and an 80 W resistojet with xenon propellant producing 220 mN of thrust at an I_{sp} of 50 s.¹¹ Both of these devices weigh approximately 200 g.¹¹

A typical resistojet, by increasing the temperature of the gas by several hundred kelvin, provides up to a 70% increase in the I_{sp} value relative to a cold gas jet.³⁸ Even in that case, however, the I_{sp} value is still in a relatively inefficient range for producing large ΔV values in a microsatellite. For example, Gibbon, et al.



Figure 8. (A): Cut-away schematic of 15 W ALSAT-1 resistojets. (B): Flight model of ALSAT-1 thruster. Figures taken from Gibbon, et al.³⁸

describe tests with various resistojets at power ranges from 10–55 W operating on xenon; these thrusters only provide an I_{sp} in the range of 40–60 s with thrust up to 90 mN.³⁸ Switching to nitrogen, the I_{sp} approached 100 s with thrust still below 100 mN.³⁸

Additionally, the thrust of a small-scale resistojets seems to underperform relative to the capabilities of a cold gas jet, without a significant savings in propellant mass (via I_{sp}). It is postulated that when power is limited by microsatellite capabilities, the mass flow must be restricted considerably to allow enough heating of the gas via the resistojets heaters. This limits the thrust that can be produced by a resistojets. When response time is a concern, a 1–3 N cold gas thruster may be a better option than a resistojets that produces 10% of the thrust at only a slightly higher I_{sp} , while requiring additional power supplies and other components.

Microscale arcjets have also been investigated for microsatellite applications. Thrust obtained by a laser machined micro-arcjet was 1.4 mN.³⁹ Although this is a relatively low thrust for a microscale electrothermal thruster, it was obtained using only 3.6 W of input power. This microscale arcjet has a specific impulse of 138 s with a thrust efficiency of 24% using a nitrogen propellant. An advantage of microscale electrothermal propulsion systems is the ability to vary the input power and propellant mass flow to meet a wide variety of mission requirements. Thrust and specific impulse can be traded in a given propulsion system to meet the needs of a particular scenario. Thruster lifetime and heat transfer issues are obvious drawbacks to microscale arcjets. These issues can be compounded at the microscale.

IV.F. Miniaturized Electrostatic Thrusters

As with resistojets, the ultimate capabilities of an electrostatic (ion engine or Hall thruster) system will be limited by the power available onboard a microsatellite. Whereas such devices offer a high I_{sp} , the thrust is ultimately limited by the power supply. A miniaturized ion engine known as the Miniature Xenon Ion (MiXI) thruster, for example, has been developed with up to 3200 s of specific impulse, operating on up to 50 W of power; the thrust, however, is limited to a maximum value of 1.5 mN.^{22,33,34}

Likewise, multiple miniaturized Hall thrusters have been constructed,^{29,30} including several with a novel cylindrical geometry intended to decrease the excessive wall-losses associated with small coaxial thrusters.^{27,28,31,32} Such miniaturized Hall thruster systems operate in the sub-1000 s I_{sp} range, and still only produce, at best, a few millinewtons of thrust.

Pushing the limits of microsatellite capability is the Hall thruster design of Berti, et al.²³ and Biagioni, et al.,²⁴ illustrated in Figure 9. This thruster, which consumes a nominal power of 100 W, can operate over the range of 60–160 W. The Hall thruster can produce 3–10 mN of thrust with an I_{sp} greater than 1000 s. Biagioni, et al. further specify that their thruster weighs 0.6 kg and that the power and flow control systems required would only add an additional 4 kg,²⁴ indicating that the system would fit well within the mass-budget of a microsatellite.

In light of the low thrust limitations of electrostatic thrusters, they are not suitable for a fast-response

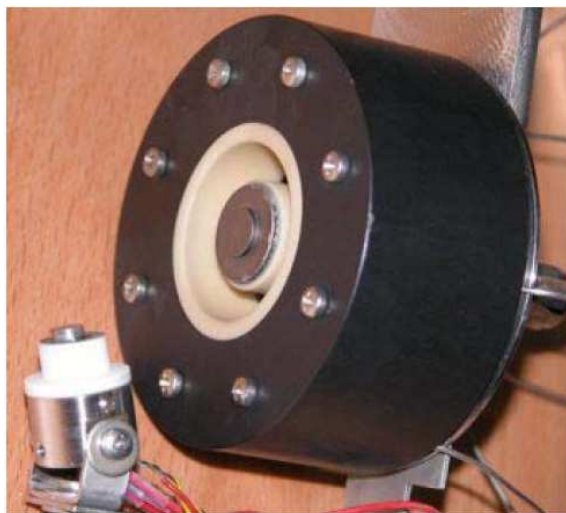


Figure 9. 100 W miniaturized Hall thruster designed to operate at 100 W nominal power, producing up to 10 mN of thrust with a maximum I_{sp} of 1000 s.^{23,24} Figure taken from Berti, et al.²³

mission in low Earth orbit. The high I_{sp} value of these systems provides the capability to deliver a 1.5 km/s velocity increment to a microsatellite with 10–20 kg of propellant. However, this would require at least 100 days of thruster operation. Provided that suitable power was available, and that time response was not an issue, an electrostatic thruster might be ideal for the scheduled maintenance or refueling of several satellites, to occur over the course of many months to several years.

IV.G. Other Noteworthy Technologies

A thruster design termed a “micronozzle with decomposing solid”⁴⁵ may also be of interest for this analysis. For this thruster type, only I_{sp} data (230 s) is currently published, but the technology offers the advantage of long-term, high-density storage.⁴⁵ This could be suitable for a microsatellite launched into a LEO parking orbit, then hibernating for years or decades until its inspection capabilities are needed.

V. Discussion

An additional note worth making is that the ultimate lifetime or propellant throughput capability of a microthruster does not often accompany its other performance data in the literature. While lifetime goals of at least several hours of operation are listed for some miniaturized chemical thrusters,²⁵ actual lifetime test data is not available for most of the thruster designs. Whereas simple cold gas thrusters, solar thermal, electrothermal, and electrostatic systems can generally be expected to run for thousands of hours, some questions remain as to whether certain MEMS-based chemical microthruster devices could provide a propellant throughput of several 10’s of kilograms.

Another note on miniaturized chemical thrusters, however, is that the listed thrust values in no way represent a maximum for this technology – clearly, there exist monopropellant and bipropellant engines with many orders of magnitude greater thrust, and it is plausible that the microrocket versions could be scaled up slightly to produce higher thrust and still remain within the mass and power limits for a microsatellite. Other technologies (i.e. electrothermal or electrostatic devices that require electric power to generate or enhance thrust) may not be able to scale up and still remain within the microsatellite power budget limits.

Nonetheless, the state of the art appears promising for microsatellites with rendezvous or inspection capabilities. Relatively high thrust miniaturized monopropellant and bipropellant chemical thrusters exist that should provide fast response times with ΔV capabilities over 1 km/s. Solar thermal systems designed for microsatellites provide moderate thrust at a high I_{sp} level that could provide for an even greater ΔV capability, but with some extra penalty in terms of the mass required to collect, concentrate, and direct

the sunlight. Miniaturized electrostatic thrusters, with I_{sp} values well over 1000 s, could provide for total velocity increments approaching 10 km/s on a microsatellite system, albeit at very low thrust levels yielding likely response times on the order of several months.

VI. Conclusion

Ultimately, for the Department of Defense and other parties that may be interested in the type of rendezvous or inspection mission discussed here, a relatively fast response time will likely be a strong deciding factor in selecting the propulsion technology for such a microsatellite.

With this in mind, the miniaturized monopropellant and bipropellant engines estimated to produce several newtons of thrust are likely ideal. With these systems, a ΔV over 1 km/s would be achieved in roughly an hour. The small size and low power consumption of these systems also allows for the option to cluster multiple thrusters to further increase a microsatellite's capabilities and flexibility. However, these miniaturized chemical rockets are yet to be exhaustively tested to precisely measure critical characteristics such as thrust, total propellant throughput, and thruster lifetime. Acceptable capabilities and longevity would need to be experimentally verified before the technology could be flown on-board a critical microsatellite mission.

Proposed solar-thermal propulsion systems can exceed the I_{sp} of a chemical system, with a relatively small sacrifice in system thrust and response time. Potentially capable of a 1.5 km/s ΔV in less than one day, rendezvous could still be achieved on a relatively short time scale. While the robustness and reliability of such a simple system is likely high, the development of these thrusters specifically for microsatellites, including options such as fiber-optic coupling and thermal storage, appears to be at a relatively early stage of development. Further research and development of this technology is recommended, as solar-thermal propulsion is another strong candidate for microsatellite-inspector systems.

References

- ¹Eshel, D., "Small Satellites Provide Low-Cost Entree," *Aviation Week & Space Technology*, April 2009.
- ²Jason, S., da Silva Curiel, A., Gomes, L., Phipps, A., Ward, J., Sun, W., and Sweeting, M., "Low Cost Planetary Exploration: Surrey Lunar Minisatellite and Interplanetary Platform Missions," *Acta Astronautica*, Vol. 48, No. 5-12, 2001, pp. 669-680.
- ³Spores, R. A. and Birkan, M., "Overview of USAF Electric Propulsion Program," *38th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2002, AIAA-2002-3558.
- ⁴Kennedy, F. G. and Palmer, P. L., "Preliminary Design of a Micro-Scale Solar Thermal Propulsion System," *38th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2002, AIAA-2002-3928.
- ⁵Walker, R., Wells, N., Green, S., and Ball, A., "The SIMONE Mission: Low-Cost Exploration of the Diverse NEO Population via Rendezvous with Microsatellites," *54th International Astronautical Congress*, The International Astronautical Federation, the International Academy of Astronautics, and the International Institute of Space Law, Sept. 2003, IAC-03-Q.5.05.
- ⁶Baker, A. M., da Silva Curiel, A., Schaffner, J., and Sweeting, M., "Advanced Low Cost Propulsion Concepts for Small Satellites Beyond LEO," *55th International Astronautical Congress*, The International Astronautical Federation, 2004, IAC-04-IAF-S.1.08.
- ⁷Walker, R., Wells, N., Green, S., and Ball, A., "The SIMONE Mission: Close Reconnaissance of the Diverse NEO Population as a Precursor to Impact Mitigation," *2004 Planetary Defense Conference: Protecting Earth from Asteroids*, American Institute of Aeronautics and Astronautics, Feb. 2004, AIAA-2004-1483.
- ⁸Kennedy, III, F. G., Palmer, P., and Paul, M., "Results of a Microscale Solar Thermal Engine Ground Test Campaign at the Surrey Space Centre," *40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2004, AIAA-2004-4137.
- ⁹Nakamura, T., Sullivan, D., McClanahan, J. A., Shoji, J. M., Partch, R., and Quinn, S., "Solar Thermal Propulsion for Small Spacecraft," *40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2004, AIAA-2004-4138.
- ¹⁰Nakamura, T., Krech, R. H., McClanahan, J. A., Shoji, J. M., Partch, R., and Quinn, S., "Solar Thermal Propulsion for Small Spacecraft - Engineering System Development and Evaluation," *41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2005, AIAA-2005-3923.
- ¹¹Barley, S. Z., Palmer, P. L., Wallbank, J. R., and Baker, A. M., "Characterization of a Monopropellant Microthruster Catalytic bed," *41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2005, AIAA-2005-4544.
- ¹²Hoyt, R. P., "Responsive Launch of Small Spacecraft Using Reusable In-Space Tether and Air-Launch Technologies," *Space 2006*, American Institute of Aeronautics and Astronautics, Sept. 2006, AIAA-2006-7278.
- ¹³Moser, R. L., Stallard, M. J., Carlin, D., and Das, A., "Low Cost Microsatellites: Innovative Approaches to Breaking

The Cost Paradigm,” *AIAA Space 2000 Conference*, American Institute of Aeronautics and Astronautics, Sept. 2000, AIAA-2000-5195.

¹⁴Mueller, J., “Thruster Options for Microspacecraft: A Review and Evaluation of Existing Hardware and Emerging Technologies,” *33rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 1997, AIAA-1997-3058.

¹⁵Mueller, J., *Micropropulsion for Small Spacecraft, Progress in Astronautics and Aeronautics*, Vol. 187, American Institute of Aeronautics and Astronautics, Inc., Reston, Virginia, 2000, pp. 45–137.

¹⁶A. M. Baker, D. Gibbon, C. U., “Micropropulsion from SNAP to PALMSAT: When does MEMS Become the Way Forward?” *NanoTech 2002 - “At the Edge of Revolution”*, American Institute of Aeronautics and Astronautics, Sept. 2002, AIAA-2002-5759.

¹⁷Ketsdever, A. D., “Microfluidics Research in MEMS Propulsion Systems,” *41st Aerospace Sciences Meeting*, American Institute of Aeronautics and Astronautics, Jan. 2003, AIAA-2003-0783.

¹⁸Koizumi, H., Inoue, T., Arakawa, Y., and Nakano, M., “Dual Propulsive Mode Microthruster Using a Diode Laser,” *Journal of Propulsion and Power*, Vol. 21, No. 6, 2005, pp. 1133–1136.

¹⁹Marcu, B., Prueger, G., Epstein, A. H., and Jacobson, S. A., “The Commoditization of Space Propulsion: Modular Propulsion Based on MEMS Technology,” *41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2005, AIAA-2005-3650.

²⁰Sahara, H. and Shimuzu, M., “Solar Thermal Propulsion System for Microsatellite Orbit Transferring,” *40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2004, AIAA-2004-3764.

²¹Norris, S., “Commerical Applications for Microsatellites,” *AIAA Space 2001 Conference*, American Institute of Aeronautics and Astronautics, Aug. 2001, AIAA-2001-4743.

²²Mueller, J., Marrese, C., Ziemer, J., Green, A., Yang, E.-H., Mojarradi, M., Johnson, T., White, V., Bame, D., Wirz, R., Tajmar, M., Hruby, V., Gamero-Castaiio, M., Schein, J., and Reinicke, R., “JPL Micro-Thrust Propulsion Activities,” *NanoTech 2002 - “At the Edge of Revolution”*, American Institute of Aeronautics and Astronautics, Sept. 2002, AIAA-2002-5714.

²³Berti, M., Biagioni, L., Cesari, U., Saverdi, M., and Andrenucci, M., “Development and Preliminary Characterization of a Low Power Hall Thruster Prototype,” *40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2004, AIAA-2004-3944.

²⁴Biagioni, L., Cesari, U., Saverdi, M., and Andrenucci, M., “Development Status of the HT-100 Miniaturized Hall Effect Thruster System,” *41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2005, AIAA-2005-3875.

²⁵Scharlemann, C. and Tajmar, M., “Development of Propulsion Means for Microsatellites,” *43rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2007, AIAA-2007-5184.

²⁶Stein, W. B., Alexeenko, A. A., and Hrbud, I., “Performance Modeling of a Coaxial Radio-Frequency Gas-Discharge Microthruster,” *Journal of Propulsion and Power*, Vol. 24, No. 5, 2008, pp. 1007–1017.

²⁷Raitses, Y., Fisch, N. J., Ertmer, K., and Burlingame, C. A., “A Study of Cylindrical Hall Thruster for Low Power Space Applications,” *36th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2000, AIAA-2000-3421.

²⁸Smirnov, A., Raitses, Y., and Fisch, N. J., “Performance Studies of Miniaturized Cylindrical and annular Hall Thrusters,” *38th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2007, AIAA-2007-3823.

²⁹Ito, T., Gascon, N., Crawford, W. S., and Cappelli, M. A., “Further Development of a Micro Hall Thruster,” *42nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2006, AIAA-2006-4495.

³⁰Ito, T., Gascon, N., Crawford, W. S., and Cappelli, M. A., “Experimental Characterization of a Micro-Hall Thruster,” *Journal of Propulsion and Power*, Vol. 23, No. 6, 2007, pp. 1068–1074.

³¹Raitses, Y., Smirnov, A., Granstedt, E., and Fisch, N. J., “Optimization of Cylindrical Hall Thrusters,” *43rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2007, AIAA-2007-5204.

³²Diamant, K. D., Pollard, J. E., Raitses, Y., and Fisch, N. J., “Low Power Cylindrical Hall Thruster Performance and Plume Properties,” *44th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2008, AIAA-2008-4998.

³³Wirz, R., Goebel, D., Marrese, C., and Mueller, J., “Development of Cathode Technologies for a Miniature Ion Thruster,” *39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2003, AIAA-2003-4722.

³⁴Wirz, R. E., *Discharge Plasma Processes of Ring-Cusp Ion Thrusters*, Ph.D. thesis, California Institute of Technology, Pasadena, California, 2005.

³⁵Simon, D. H. and Land, H. B., “Pulsed Plasma Thrusters for Micropropulsion,” *39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2003, AIAA-2003-5170.

³⁶Bayt, R. and Breuer, K., “Systems Design and Performance of Hot and Cold Supersonic Microjets,” *39th AIAA Aerospace Sciences Meeting*, American Institute of Aeronautics and Astronautics, Jan. 2001, AIAA-2001-0721.

³⁷Kappenstein, C., Bottonneau, Y., and Ford, M., “Chemical micropropulsion. State of the art and catalyst surface requirements,” *41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2005, AIAA-2005-3920.

- ³⁸Gibbon, D., Baker, A., Nicolini, D., Robertson, D., and Dye, C., "The design, development and in-flight performance of a low power resistojet thruster," *39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2003, AIAA-2003-4548.
- ³⁹Horisawa, H., Onodera, K., Noda, T., and Kimura, I., "Optimum operational conditions and microfabrications with UV lasers of very low-power DC micro-arcjets," *Vacuum*, Vol. 80, 2006, pp. 1244–1251.
- ⁴⁰Bzibziak, R., "Update of Cold Gas Propulsion at Moog," *36th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2000, AIAA-2000-3718.
- ⁴¹Darnon, F., LeDeuff, Y., and Pillet, N., "DEMETER and PARASOL Micro-satellites: Flight Performances of the 1N Hydrazine Propulsion System," *41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2005, AIAA-2005-3953.
- ⁴²An, S., Lim, H., and Kwon, S., "Hydrogen Peroxide Thruster Module for Microsatellites with Platinum Supported by Alumina Catalyst," *43rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2007, AIAA-2007-5467.
- ⁴³Hitt, D. L., Zakrzewski, C. M., and Thomas, M. A., "MEMS-based satellite micropropulsion via catalyzed hydrogen peroxide decomposition," *Smart Materials and Structures*, Vol. 10, 2001, pp. 1163–1175.
- ⁴⁴Volchko, S. J., Sung, C.-J., Huang, Y., and Schneider, S. J., "Catalytic Combustion of Methane/Oxygen Mixtures for Micropropulsion Applications," *41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2005, AIAA-2005-3924.
- ⁴⁵Reed, B., de Groot, W., and Dang, L., "Experimental Evaluation of Cold Flow Micro-Nozzle Performance," *37th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2001, AIAA-2001-3521.
- ⁴⁶Phipps, C., Luke, J., and Marquis, J., "Diode Laser-based Microthrusters: A New Departure in High I_{sp} , Long-Life Engines," *36th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2000, AIAA-2000-3477.
- ⁴⁷Grubisic, A. N. and Gabriel, S. B., "Hollow Cathodes as a Plasma Propulsion Device," *46th AIAA Aerospace Sciences Meeting*, American Institute of Aeronautics and Astronautics, Jan. 2008, AIAA-2008-1082.
- ⁴⁸Sahara, H. and Shimizu, M., "Solar Thermal Propulsion System for a Japanese 50kg-Class Microsatellite," *39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2003, AIAA-2003-5032.
- ⁴⁹Henshall, P. R., Palmer, P., and Baker, A. M., "Solar Thermal Propulsion Augmented with Fiber Optics: -A System Design Proposal," *41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2005, AIAA-2005-3922.
- ⁵⁰Henshall, P. and Palmer, P., "Solar Thermal Propulsion Augmented with Fiber Optics: - Technology Development," *42nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2006, AIAA-2006-4874.
- ⁵¹Mosher, T. J., Kwong, J., Florian, B., Lin, C., Mitchell, M., Quiroz, R., Ringler, S., and Segal, Y., "Analysis and Design of a Tether Boost Facility for Martian Satellite Transfer," *39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, American Institute of Aeronautics and Astronautics, July 2003, AIAA-2003-4445.
- ⁵²Yetter, R. A., Yang, V., Aksay, I. A., and Dryer, F. L., "Chemical Microthrusters: Combustion Issues and Approaches," *Proceedings of the International Symposium on Space Propulsion*, Shanghai, China, 2004.